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AUTHOR(S)- I. Silberstein

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ABSTRACT

Manned missions to features of scientific interest on the lunar surface will have to land, in many cases, on sites that are smaller and more hazardous than the Apollo sites. The guidance method planned for Apollo may not be sufficiently versatile nor accurate enough to perform landings at many of these points.

A landing procedure that may make such landings possible is proposed. It is based on complete position vector updates during the braking maneuver using landmark identification. Minor changes in hardware and software are expected. Additional data are necessary before the feasibility of this method is reasonably verified.

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SUBJECT: LM Landing Using Steep Descent Trajectory - Case 340

DATE: April 3, 1968

FROM: I. Silberstein

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TECHNICAL MEMORANDUM

INTRODUCTION

The problem of landing a Lunar Module (LM) to a point on the Moon is investigated. The object is to design a landing method to a predetermined point in hazardous areas on the Moon. The hazards impose essentially two constraints on the landing. First, a high degree of accuracy in landing must be achieved so that the LM may avoid neighboring points where landing is too dangerous. Second, the trajectory must be relatively independent of the terrain so as to avoid violent changes of pitch resulting from terrain elevation difference. The landing procedure proposed below may make such landings possible with only minor modifications of hardware and software and without calling for more accurate mapping and guidance than is already attainable.

The visibility phase of the trajectory planned for Apollo missions is comparatively flat (Fig. 1). High gate is approximately 30,000 ft uprange and 8,500 ft above the landing point. During the braking phase, starting at about 35,000 ft altitude, the LM flys windows up and the only interaction of the LM guidance system with the lunar surface is through the landing radar (LR), which supplies the guidance system with altitude and velocity updates. The landing area and the approach ray are selected so that the terrain is smooth and does not cause the LM to execute excessive pitch maneuvers. The landing area used for site selection purposes is approximately the same size as the 3σ error ellipse of the LM guidance system and the LM is free to land anywhere within that area (Fig. 2). The only restriction is that the LM must avoid obstacles larger than a certain size.

TECHNICAL DISCUSSION

The landing point in "science sites" may be surrounded by rough terrain. In this case it is desirable to provide independence of the trajectory from the terrain, both before and after high gate. After high gate this goal may be accomplished by pointing the landing radar at the landing site as soon as the landing site is acquired with the Landing Point Designator (LPD). If the approach trajectory after high gate is too flat, three adverse conditions may result. First, the dispersed range beam will cover an area too large for an accurate range measurement. Second, the distance from the LM to the target will be too large for adequate radar operation and, third, the angle of incidence of the beam at the lunar surface will not be favorable for adequate power return of the reflected signal. Steep trajectory solves these problems since, for a given time from high gate to landing, the range is shorter and the angle of incidence is more favorable.

In addition to the influence of the terrain, considerations of visibility constraints enter into the trajectory design. The visibility requirements imply that during the final approach phase the trajectory must be steeper than the sun elevation angle. During later lunar landings to science sites it will be necessary to land essentially to a point. Landing to a point implies that the nominal trajectory should be biased uprange so that, in no case, will redesignations uprange be required. (The resulting pitch variations cause loss of visibility.) Biasing the trajectory in such a manner flattens the approach and reduces visibility. A 3σ uprange error from the biased landing point flattens the trajectory even further (Figure 3). Thus, landing could be attempted only when the sun is less than 8° above the local horizon.

In order to insure a high launch probability during any given month, sufficient time for two countdown recycles, that is, a 5 day launch window is desirable. However, under current constraints, acceptable lighting conditions at a given point on the Moon (near the equator) last for, at most, one day. Two complementary steps may be taken; the first is to schedule the first launch opportunity a few days before lighting conditions are appropriate and modify the flight time, so that landing will occur when lighting conditions are favorable at the landing site. The second is to increase the steepness of the trajectory so that the lighting requirements may be relaxed, and an increased landing period made available.

All of the above factors lead to a consideration of steep trajectory landing. Safety requirements (avoidance of "dead man's trajectory") prohibit the use of near vertical trajectories. The LM window geometry imposes an absolute maximum of 65° on the steepness, but the practical maximum is probably 50° to 55°.

Approximate calculations of steep LM trajectories (35° and 45°) were carried out (Appendix). The preliminary data obtained for 45° landing are used in the discussion below (Figure 4).

During a steep trajectory landing, it is imperative that the LM position error be corrected before high gate. The requirement of minimizing fuel expenditure implies that the final approach phase cannot last long. Thus, the final approach for a 45° landing must start about 8,000 ft uprange from the landing point (Fig. 5). However, the present 3σ downrange error is in excess of 12,000 ft. Thus, it is quite possible that when the pitch angle changes at high gate the landing point will be behind the LM (Fig. 6). Even assuming that the pilot realizes immediately what has happened, he may not have enough fuel to change the direction of motion and conduct a safe landing.

It appears that the problem is primarily one of accurate navigation. If the position corrections are made early enough they cost less fuel. If the nominal high gate position is achieved, much of the fuel spent to shape the trajectory may be recovered, as very little fuel will be needed for redesignations.

METHOD OF IMPLEMENTATION

Two methods of achieving sufficient accuracy were examined. The first involves getting a fix of the LM from the CSM, using the CSM sextant and rendezvous radar. However, a few difficulties are encountered. First, even if the position of the LM is found accurately with respect to the CSM, the position of the CSM is known only with respect to some point on the surface which is not necessarily the landing point. The inaccuracy of the transformation between two such points is sufficient to cause large ambiguities in the location of the LM with respect to the landing point. Second, no direct transfer of data from the CSM to the LM presently exists. Third, a few updates may be needed in the final stages of the braking maneuver to correct the trajectory of the LM, and the CSM pilot may not have time to update his own position and get fixes of the LM.

The second method considered involves landmark identification by the LM crew during the final portion of the braking maneuver. The data acquired by identifying those landmarks, and the position of the LM with respect to them, may be used to correct the LM flight trajectory before high gate.

With the LM flying windows down almost all the way to high gate, the crew can use the LPD (unless something better is devised by 1972-73) for the identification and marking of the landmarks. The landing radar will be used for range measurements. The landing radar will have to be articulated and modified so that it can point to the ground when the windows are down as well as when they are up.

The LM crew will identify a landmark whose position vector, with respect to the high gate position of the LM, will have been previously stored in the LM guidance computer (LGC). The landmark, in general, will be outside the plane of the trajectory. The LM pilot will execute a body roll (pilot yaw) and superimpose the center line of the LPD on the landmark (the landing radar rolls with the rest of the LM). The yaw registers in the LGC automatically through the inertial measurement unit (IMU). When the landmark passes under a predetermined angle mark on the LPD (say 10°) the radar range beam points at the landmark, and the LM pilot causes the data stored in the LGC to enter the calculation cycle and the landing radar range data to be sampled. The computer first calculates the position vector of the LM with respect to the landmark and then adds it to the position vector from high gate to the landmark. The result is a value for the position vector of the LM with respect to high gate, based on the landmark identification. We will refer to it as the "Landmark Position Vector".

The usual 2 second computation cycle in the computer yields another position vector based on integration of the equations of motion subject to input from the IMU accelerometers. We will refer to it as the "IMU Position Vector". The two position vectors in general will not agree, and both are in error. However, the position vector obtained from the landmark identification is more accurate (subject to current error estimates of the IMU Position Vector) <u>if the landmark is near enough to high gate</u>. The IMU Position Vector in the LGC must be updated so that it may then be used in the guidance equations.

In the current guidance program, the difference in altitude between the altitude component of the IMU position vector and the radar altitude measurement is multiplied by a linearized weighting function, then added to the calculated value. The same strategy may be used for the new position vector found by the landmark identification process.

After the update has been completed, the radar is not sampled until the next landmark. The computer uses the last updated position vector and the IMU output to guide the LM to high gate until a new update for the position vector becomes available at the next landmark and the process is repeated.

The updates will start at about 30,000 ft altitude and about 250,000 ft uprange (for a 45° landing) from high gate. The LM is in that position about 200 seconds prior to high gate. The last landmark update should be approximately 30 seconds before high gate, that is, approximately 10,000 ft uprange from high gate. During these 170 seconds it is expected that the crew may be able to identify four or five landmarks.

In the beginning of the identification process, two or three landmarks should be considered across the width of the trace of the error ellipsoid (that is, orthogonal to the nominal trajectory plane). This is necessary to minimize the yaw maneuvers and subsequent reduction of the landing radar performance. The quadratic guidance brings the vehicle quickly to its nominal plane if "S curve" maneuvers are allowed, and after one or two landmarks are identified, the yaw maneuver will be very small.

LANDMARK IDENTIFICATION

The landmarks should be relatively flat areas in the immediate vicinity of easily identifiable features, large enough to insure quick identification. The flatness requirement is due to the 6° dispersion of the landing radar range beam. The landmark should be as near as possible to the nominal trajectory plane to minimize yaw maneuvers.

A simple navigation aid could be incorporated to make the landmark identification easier. It would consist of a pitch angle display and a chart with time marks moved by a clock. The chart, with a continuously variable mapping scale,[1] will display the area under the LM as the LM moves in its nominal trajectory. In particular, the chart will display the landmarks to be identified and the nominal time at which the LM is expected to be above them. The time marks on the chart and the display of the nominal time at which the landmark is expected under the LM serve a double function: (1) once the first landmark has been identified the pilot should know when to expect the next one; (2) any large discrepancy between the actual elapsed time from one landmark to another, and the nominal time would be an indication of a failure in the navigation system. The pilot will be expected to compare the chart to the terrain below for approximately a minute prior to the first landmark. This procedure will enable him to anticipate the landmark, thus aiding the identification. This device will also free the second crew member from navigational tasks, enabling him to monitor the LM systems and perform other tasks.

The time needed for the identification and marking of a prominent feature on the lunar surface was estimated as 30 seconds. No realistic estimate can be made until simulations are carried out. The minimum available time for identification is 25 seconds, which is the time needed for the LM to advance from the point where the first landmark can be seen through the window to the point directly above the landmark.

HARDWARE IMPLICATIONS

The procedure described above is impossible unless some modifications are made to the landing radar antenna and associated logic. First, the position of the antenna must be changed. Two problems result from its present location: (1) the radar beam may be reflected off the descent engine skirt at windows down position, and (2) interference of the beam with the landing gear is also possible at certain antenna angles (Figs. 7 and 8). To satisfy the first requirement the antenna will have to be moved nearer to the landing gear which may aggravate the second problem, according to R. Harrington[2] of Ryan Aeronautical Co. He indicated that the position will have to be determined experimentally and that the experimental facility exists at the Ryan plant.

Second, structural problems may arise as a result of articulation and the change of location. These problems will have to be investigated.

Third, a 180° rotation of the LM to windows down position would cause the radar beam configuration to be inverted. In order to remedy this problem the LR antenna must also be rotated 180°. This would cause a sign change in the down and cross range components of the velocity vector. A simple change in the logic could cause these components to have the right sign. All that may be needed is for the computer to multiply them by -1 after high gate (Figs. 9 and 10).

The configuration change may actually be advantageous to the radar performance. During the second phase after high gate, the range beam will be expected to point at the landing site. With the Apollo configuration and a line-of-sight of 45°, the third velocity beam would be almost parallel to the surface. With the modified configuration the third velocity beam is almost perpendicular to the surface.

SOFTWARE IMPLICATIONS

Some modifications in the software would also be necessary. First, altitude updates will not be made continuously to high gate. Updates will be made only at identified landmarks, whose position vectors are known with respect to high gate. The position vector of the LM will be updated in all three coordinates rather than in altitude only. The existing guidance program does precisely that, but only one component of the correction is calculated from the landing radar measurement. Therefore, only minor changes may be necessary in the program itself. Weighting functions which would minimize fuel expenditure while maximizing the accuracy should be developed.

Second, logic for the articulation of the radar antenna will be needed. Three modes are necessary:

1. During the braking phase the range beam must point at a fixed angle with respect to the thrust axis. No articulation is needed.

2. During the second phase (after high gate) the range beam must point to the landing site. The LM guidance computer continuously calculates the angle to the landing site and displays the value for redesignation purposes. The same value could be used as a servo input to point the range beam to the landing site.

3. During the vertical final approach, the servo input may be the pitch angle.

ASSOCIATED ERRORS

In general, the ambiguity of the position vector of the LM calculated from the landmarks is due to three types of errors: that of the landmark position with respect to high gate, that of the LM with respect to the landmark, and that of the guidance system. The causes of these errors are:

1. Mapping errors; relative height and position of points on the lunar surface

2. LPD errors in identifying the landmark

3. Landmark ambiguity

4. Landing radar errors

5. Time delay from marking to data sampling (pilot reaction time)

The ambiguity of distance between two points on contiguous, medium resolution Lunar Orbiter V photographs is, at most, 800 ft[3]. Relative elevation ambiguities are rather large since general slopes up to 2% may not be detected. However, those elevation ambiguities could not exceed 3,000 ft. The uncertainty in the relative height diminishes in proportion to the distance between two points, and if the last landmark identified is 10,000 ft uprange from high gate, the relative height uncertainty is no more than 200 ft, a tolerable amount. Plane change maneuvers needed to correct for cross range errors will be made early, insuring that the LM is very nearly in the nominal trajectory plane before the last two or three landmarks are identified.

F. Heap(4) estimates that LPD errors are between .5° and 2°. The resulting uncertainty in the location of the landmark is a function of the range from the LM to the landmark. That range is largest at the first landmark and smallest at the last. Correspondingly, the position error associated with the LPD inaccuracy is less than 900 ft at the first landmark and less than 300 ft at the last one. This error is in both cross range and uprange directions.

By landmark ambiguity we mean the difficulty in deciding which point on the feature is the one the pilot should use in the marking process. The landmark must be a fairly large feature to make the identification process easy. It is difficult to estimate the position error resulting from this ambiguity but it seems reasonable that 2° resolution will be possible. The resulting position error can therefore be estimated to be the same as the one associated with LPD errors.

The landing radar range error is 1.5%(5), adding to the difficulty of height estimation. A small amount of coupling with the cross range may be removed in early stages as previously explained. The altitude error varies from about 500 ft at 35,000 ft altitude to 200 ft at 13,000 ft altitude (30 seconds before high gate).

The pilot reaction time may be assumed as 1/10th of a second, causing a range error of 250 ft at the first landmark and 15 ft at the last landmark. This error is negligible since the pilot anticipates the passage of the point under the LM and therefore his reaction time is very short(6).

Adding all the possible errors at high gate and assuming that the guidance system guides the LM to high gate as computed at the last landmark, it is found that the LM is within less than a thousand feet of nominal high gate. With this accuracy the landing point may be identified almost immediately, say within 30 seconds. For the 45° landing, 1,000 ft redesignation after high gate is estimated to cost less than 20 ft/sec.

The tentative choice of a 45° landing can now be explained in terms of the error analysis and other considerations. The position errors resulting from landmark identification are proportional to the distance of the landmark from high gate. If the trajectory is appreciably flatter, the LM has a higher velocity at high gate. It is thus farther from high gate at any given time-to-go on the flat trajectory than it is on the 45° trajectory at the same time-to-go. The result is that the first landmark identification will probably occur far enough uprange to be useless. The ambiguity of the landmark location relative to high gate may be larger than the error to be corrected. On the other hand, increasing the landing angle will not improve the accuracy of the trajectory sufficiently to justify the additional fuel expenditure. In addition, the LM will be nearer to a "dead man's" trajectory. The choice of 45° landing must be tested by simulations. If it is found that three landmark identifications are sufficient or that the identification process requires less than 30 seconds, the angle could be smaller, and vice-versa.

Although the general tasks which will have to be carried out by the crew during landing have been identified, the exact task schedule has not yet been planned.

FURTHER STUDY

The main problems to be solved to complete the feasibility study are:

1. Simulations of landmark identifications

2. Crew activity schedule re-evaluation

3. Radar articulation problems, and antenna location

Some of the doubts as to the validity of the landing procedure outlined above cannot be resolved without simulations and experiments. However, missions to scientifically interesting lunar features may impose landing conditions like those that have been described. In these cases the Apollo LM landing strategy is inadequate. The choice is, therefore, to forego the exploration of features where guidance problems are more complex or to try and modify the guidance method.

2015-IS-hjt

I. Silberstein

Attachments:
References
Figures 1 to 10
Appendix
Acknowledgment

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5. Cheatham, D. C., and F. V. Bennett, "Apollo LM Landing Strategy", Apollo Lunar Landing Mission Symposium, Vol. I, June 25-27, 1966

6. I. Saulietis, MSC, Personal communication

7. Bolt, W. M. and F. V. Bennett, "Proposed LM Powered Descent Trajectory for the Apollo Lunar Landing Mission", MSC Internal Note No. 67-FM-117, August 15, 1967

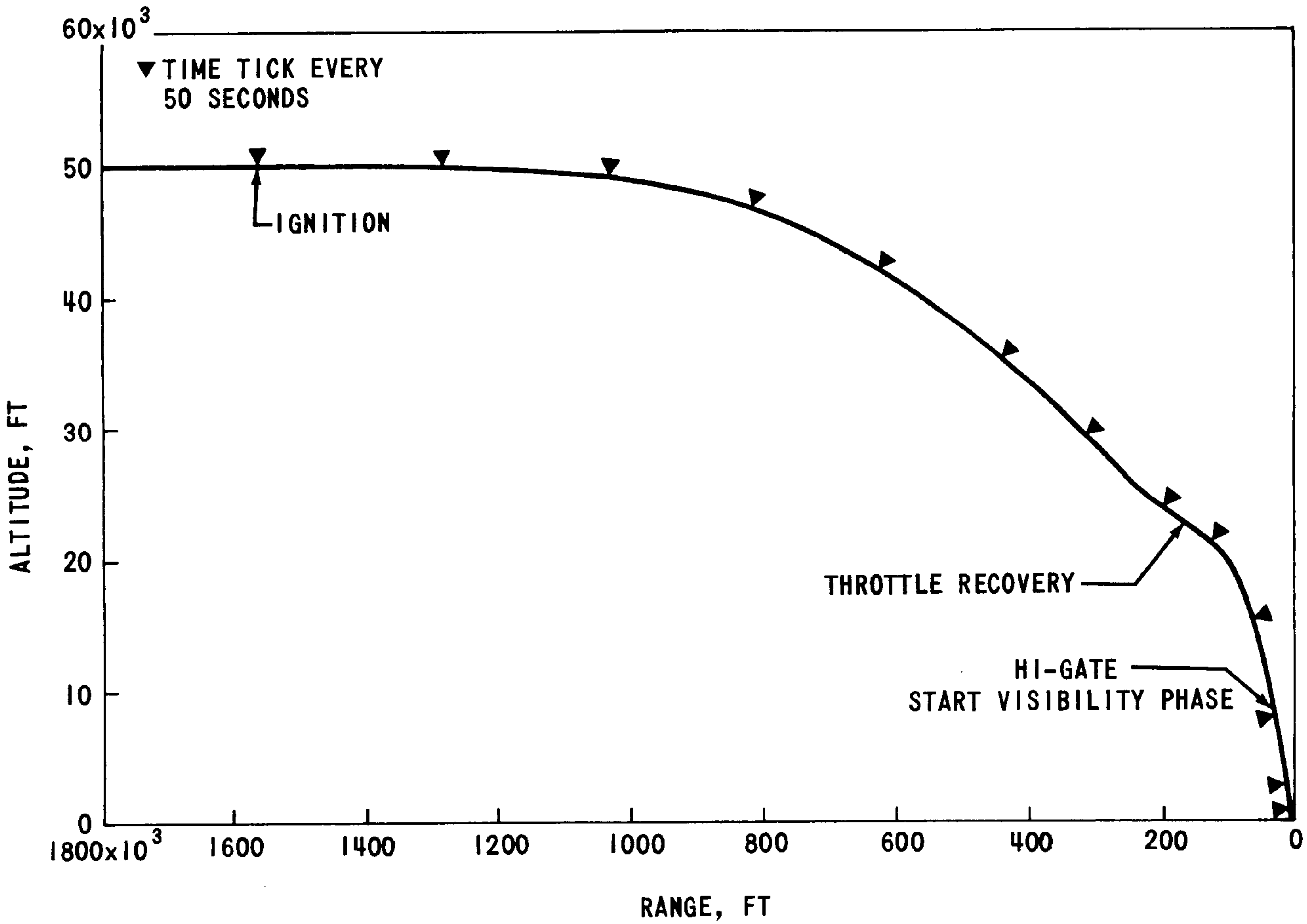


FIGURE I - ALTITUDE-RANGE PROFILE FOR LM LUNAR DESCENT TRAJECTORY

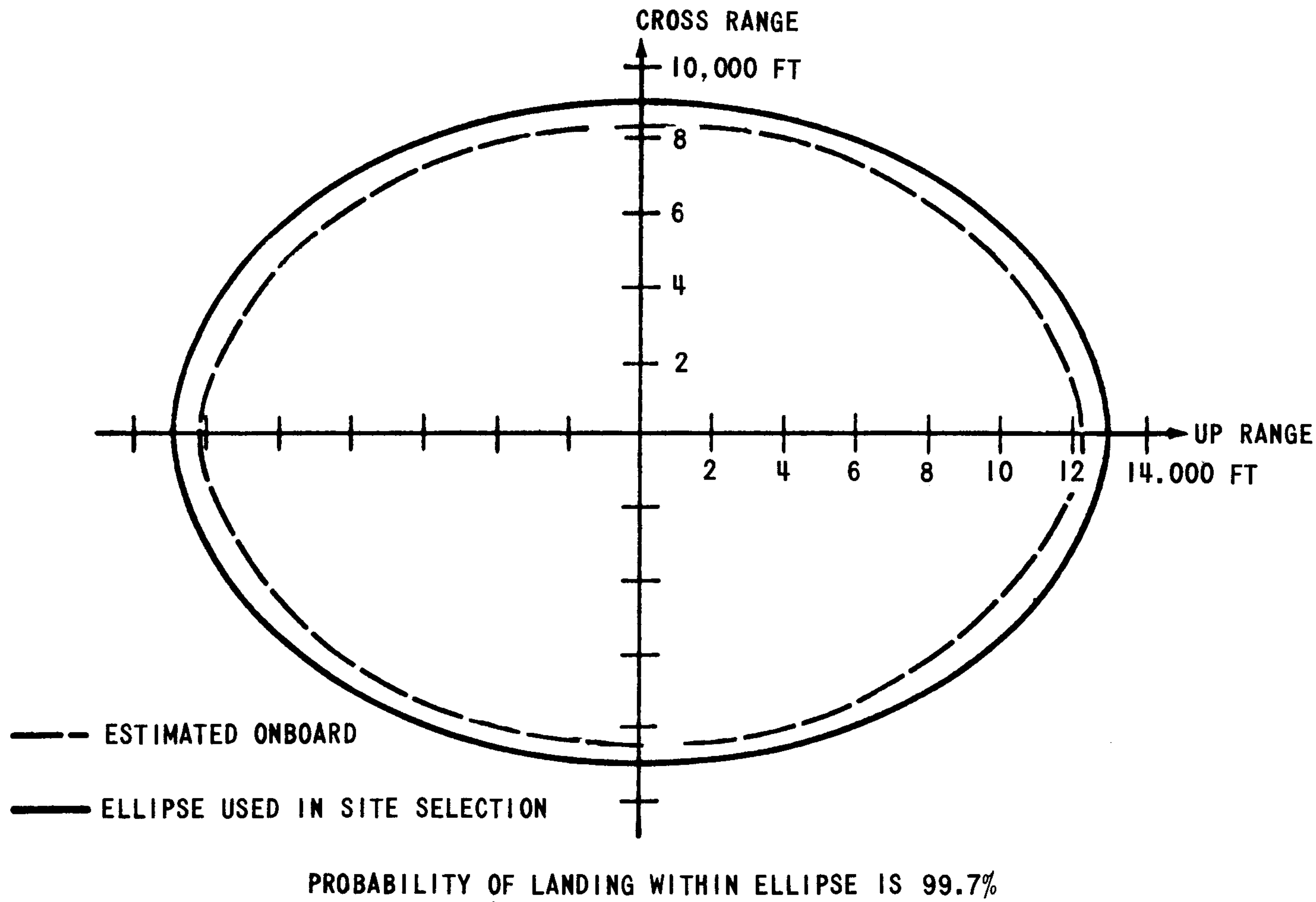
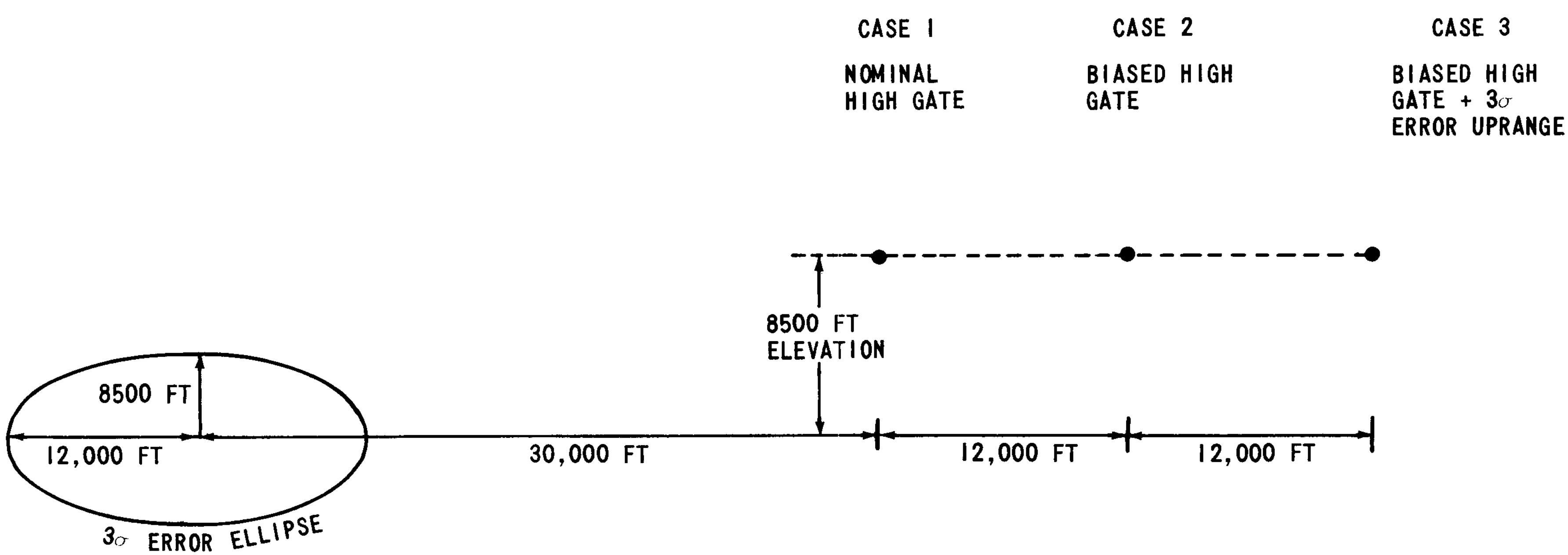


FIGURE 2 - LANDING SITE DISPERSION ELLIPSE



	CASE I	CASE 2	CASE 3
LINE OF SIGHT TO LANDING POINT AT HIGH GATE	15° +	11° +	8°

FIGURE 3

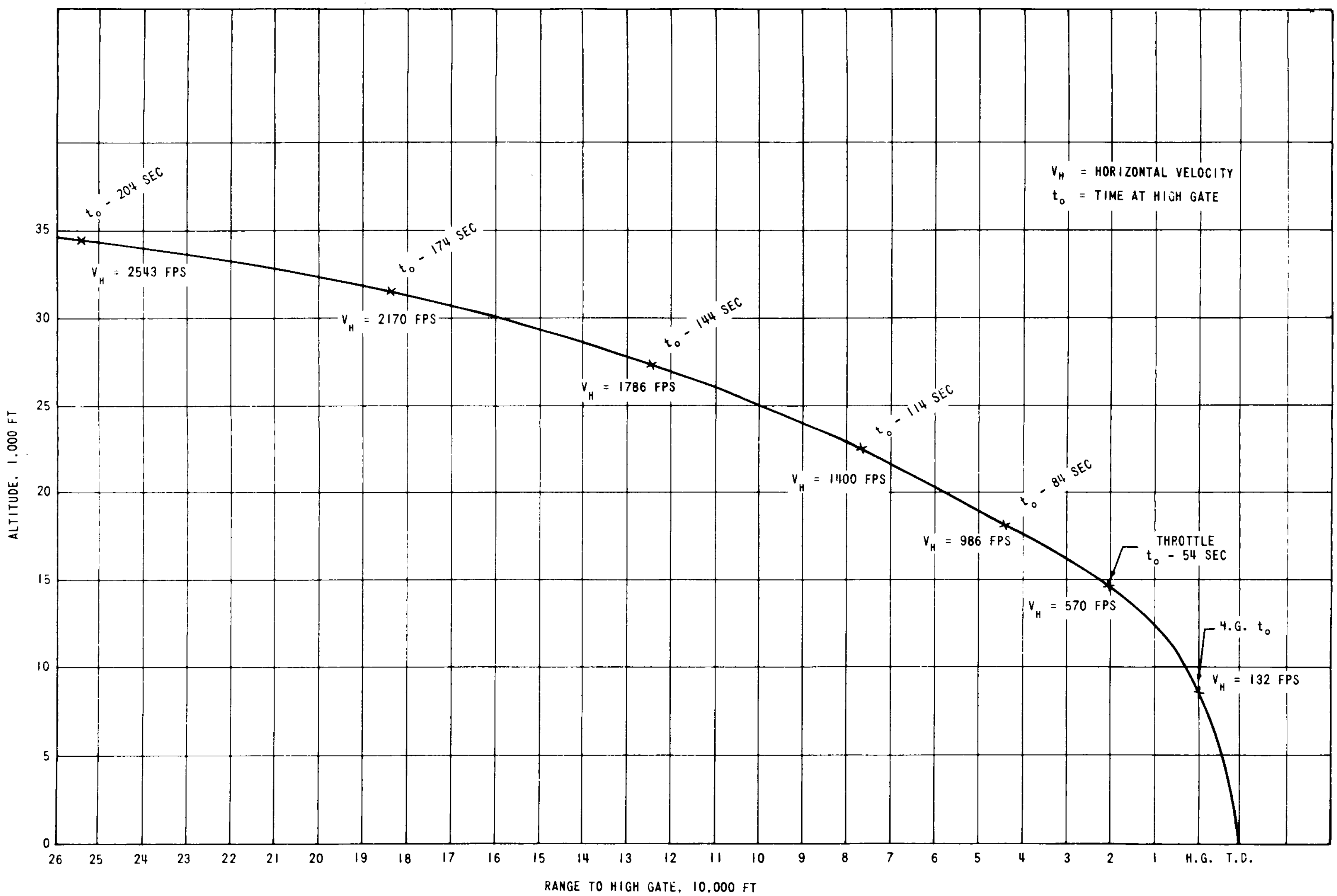
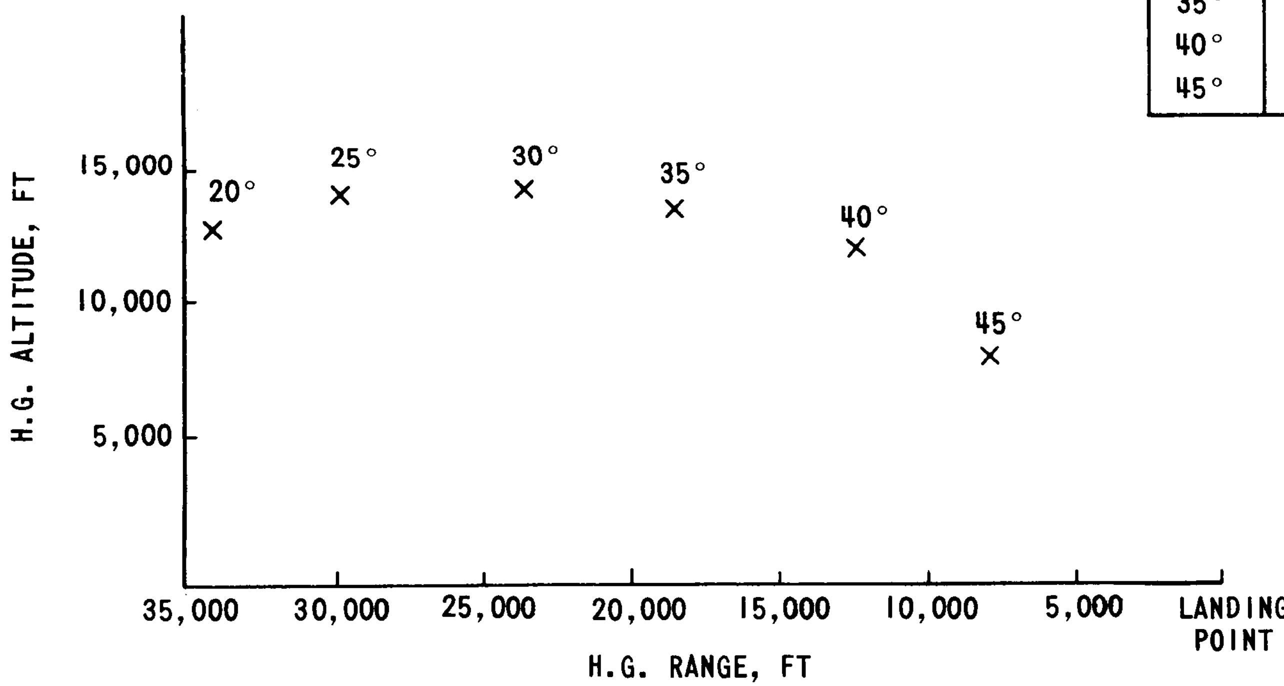


FIGURE 4 - APPROXIMATE!! APPROACH PATH FOR 45° FINAL LANDING ANGLE



	RANGE (FT)	ALT. (FT)
20°	34,500	12,600
25°	29,400	13,700
30°	23,600	13,700
35°	18,600	12,600
40°	12,800	11,400
45°	8,000	8,000

FIGURE 5 - HIGH GATE LOCUS 120 SEC FROM H.G. TO HOVER. CONSTANT PITCH ANGLE: CONSTANT TRAJECTORY SLOPE

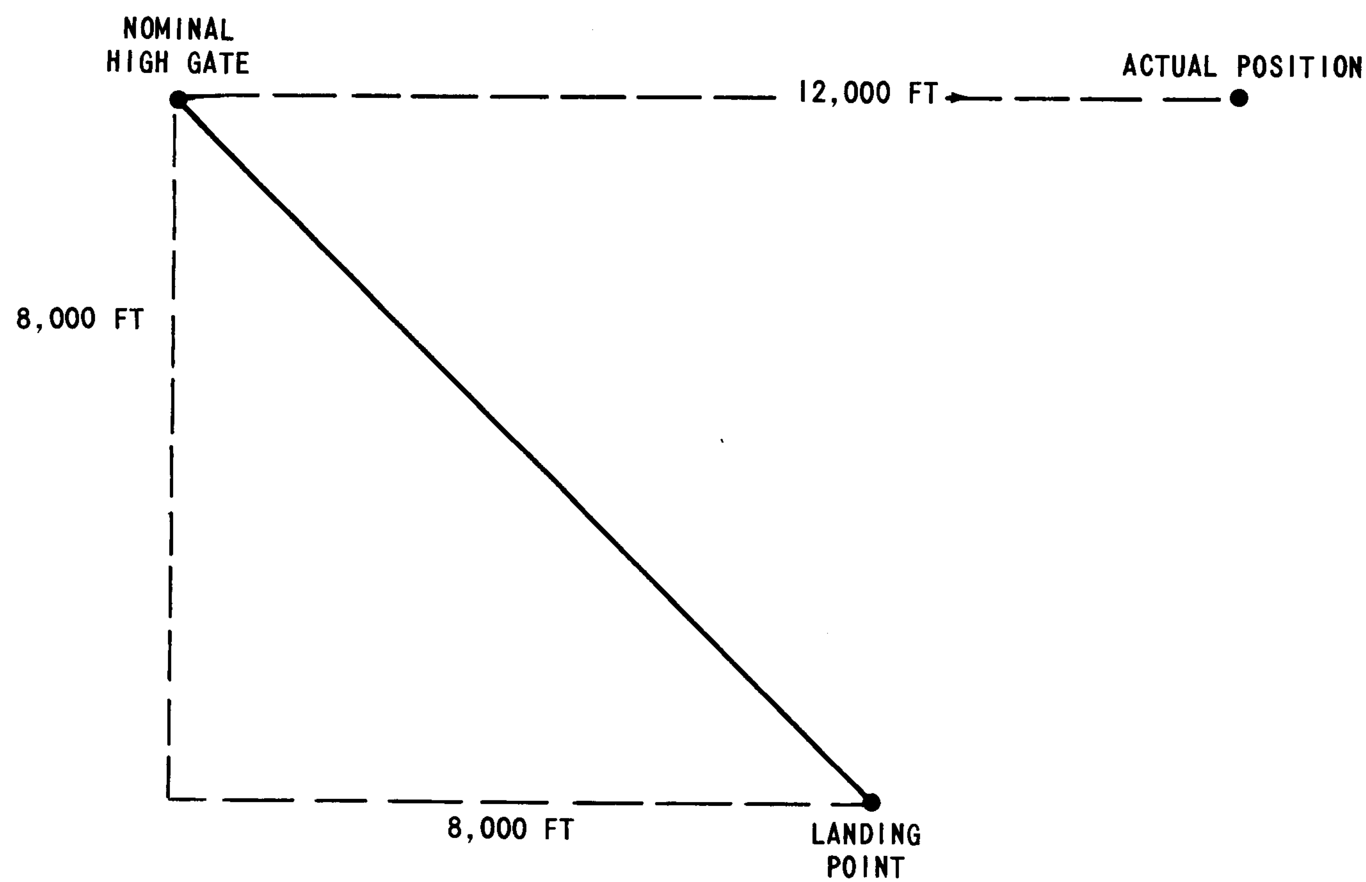


FIGURE 6 - HIGH GATE POSITION - STEEP TRAJECTORY WITH
3σ DOWNRANGE ERROR

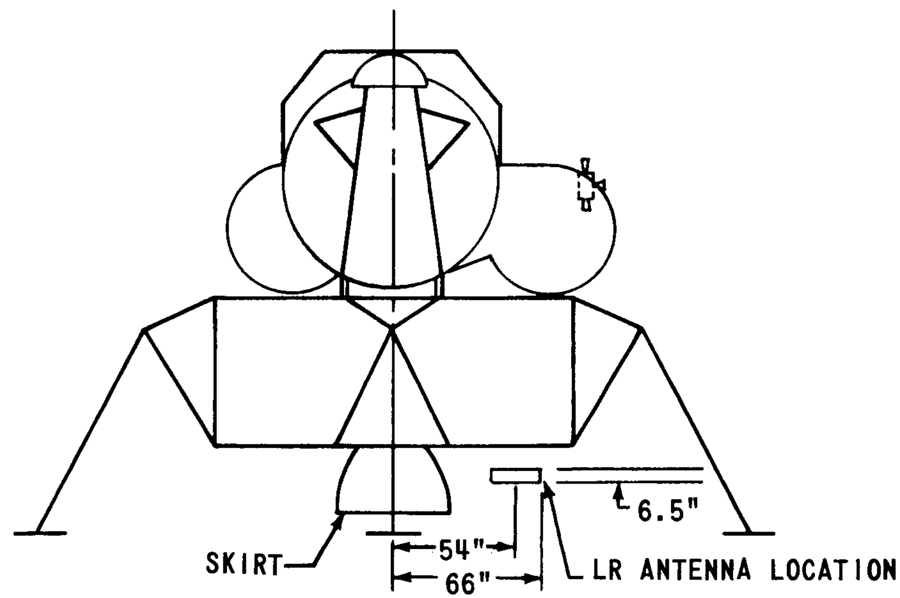


FIGURE 7 - PRESENT LANDING RADAR ANTENNA LOCATION X-Y PLANE

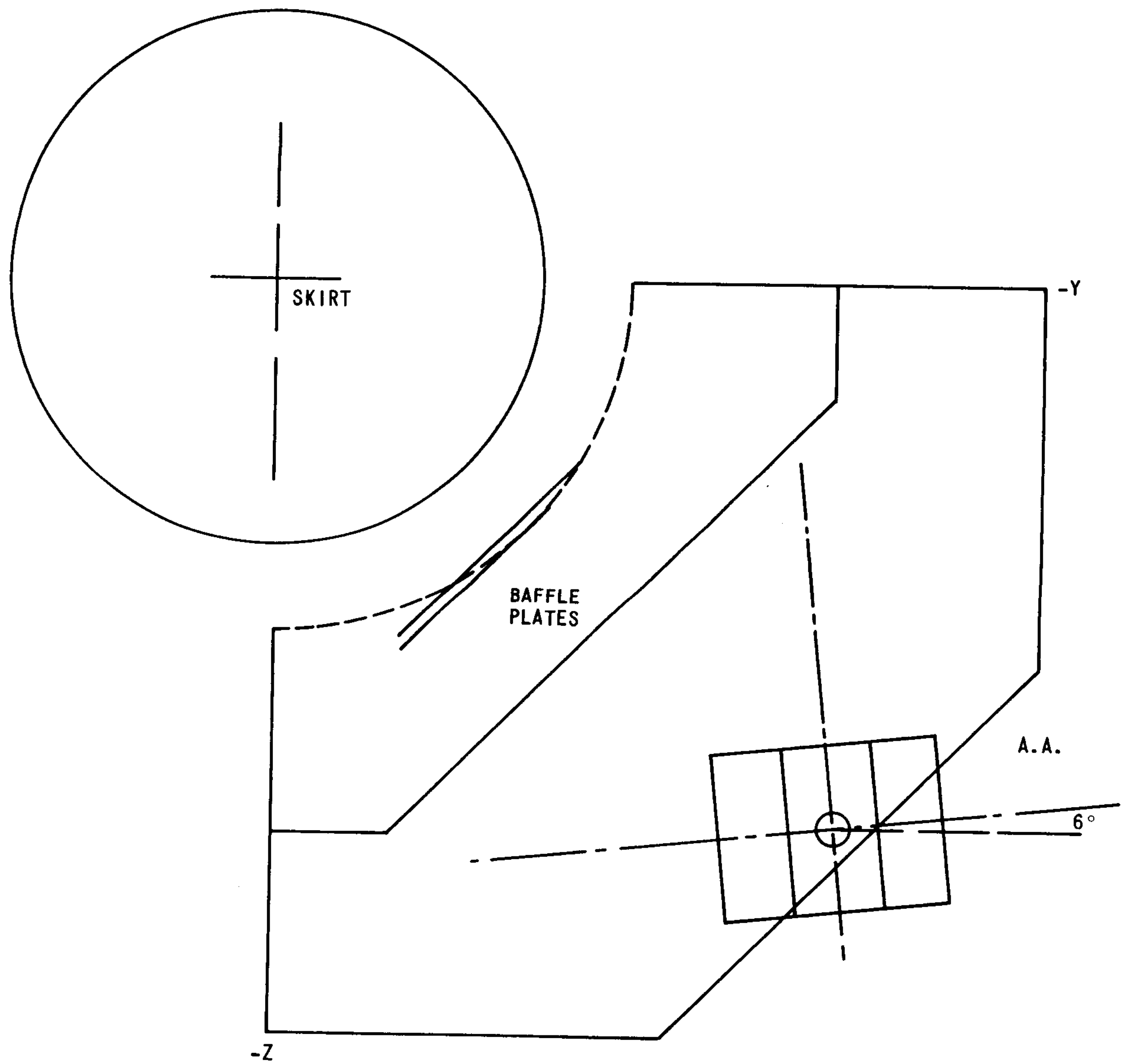


FIGURE 8 - PRESENT ANTENNA ASSEMBLY - BAFFLE PLATE LOCATION

APOLLO BRAKING PHASE-WINDOWS UP CONFIGURATION

LM INVERTED BRAKING PHASE-WINDOWS DOWN CONFIGURATION

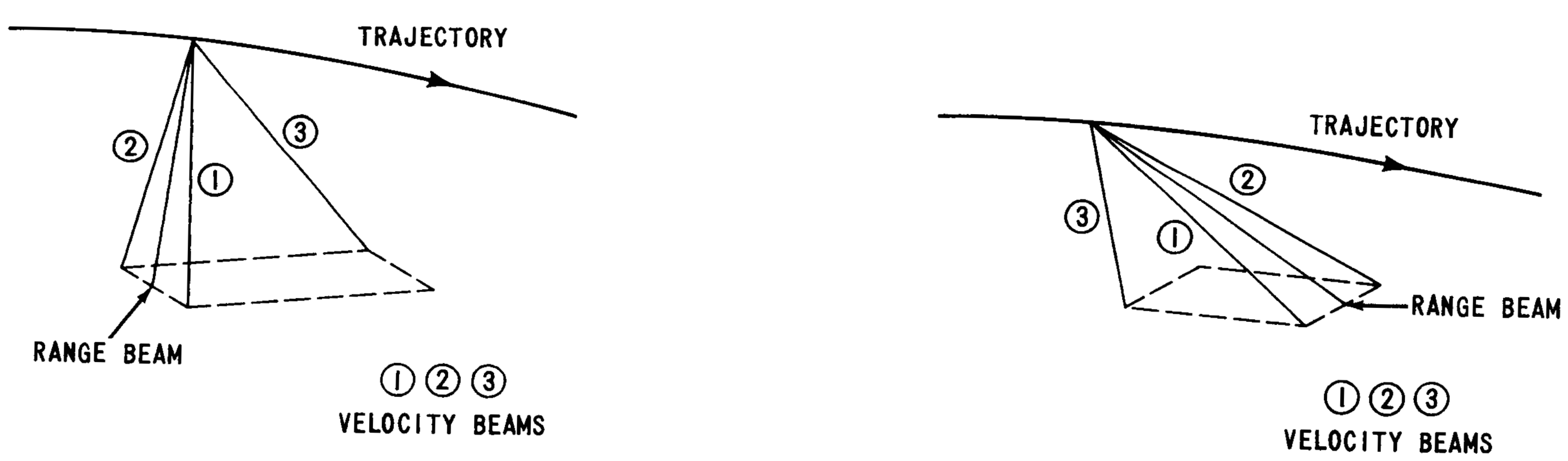


FIGURE 9 - LM INVERTED, INFLUENCE ON BEAM GEOMETRY

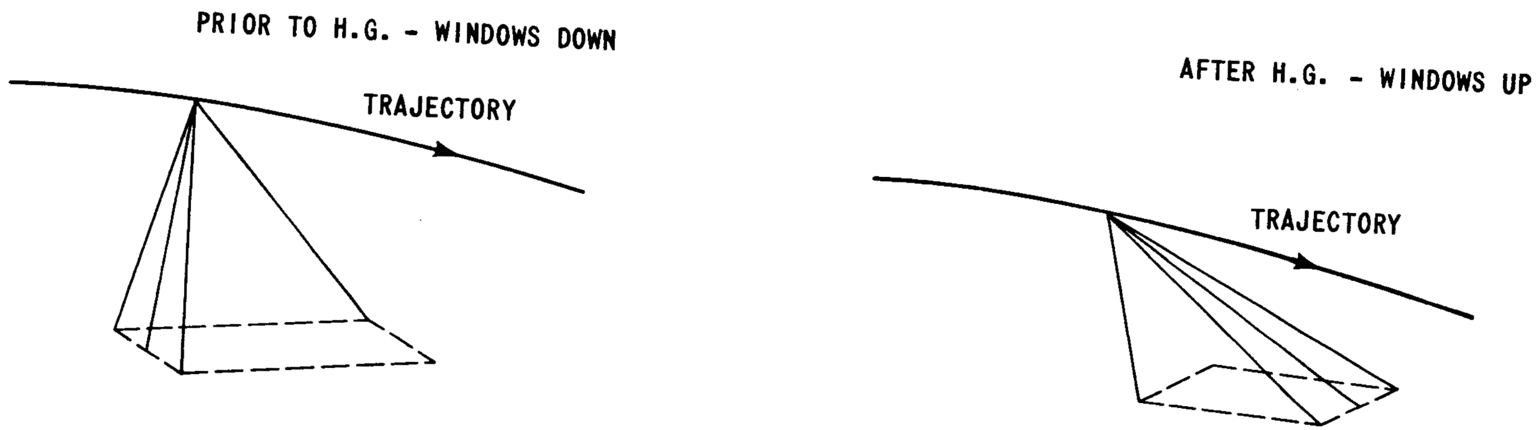


FIGURE 10 - L.R. ANTENNA INVERTED, INFLUENCE ON BEAM GEOMETRY

APPENDIX

METHOD OF APPROXIMATION OF LM DESCENT TRAJECTORY

The objective of the approximation was to obtain a rough estimate of the position and altitude of the LM up to 200 seconds prior to high gate. These data were needed to determine the feasibility of the landmark identification procedure and the steepness of the slope which will be necessary.

The basis of the calculations was a computer output obtained from F. Heap. The output included data about the state vector, the pitch and the mass of the LM at different points along the approach path. The problem was that no data were available for the 280 seconds prior to throttle recovery.

The approximation scheme included the following simplifications: (1) flat moon approximation; (2) constant pitch rate (based on the difference in pitch from throttle recovery and 280 seconds prior); (3) thirty second time intervals for the integration of the equations of motion.

The data obtained are summarized in Figure 4.

ACKNOWLEDGMENT

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